NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Space Programs Summary 37-45, Vol. VI

Space Exploration Programs and Space Sciences

For the Period March 1 to April 30, 1967

JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

PASADENA, CALIFORNIA

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Preface

The Space Programs Summary is a six-volume bimonthly publication designed to report on JPL space exploration programs and related supporting research and advanced development projects. The titles of all volumes of the Space Programs Summary are:

Vol. I. *The Lunar Program* (Confidential)

Vol. II. *The Planetary–Interplanetary Program* (Confidential)

Vol. III. *The Deep Space Network* (Unclassified)

Vol. IV. *Supporting Research and Advanced Development* (Unclassified)

Vol. V. *Supporting Research and Advanced Development* (Confidential)

Vol. VI. *Space Exploration Programs and Space Sciences* (Unclassified)

The Space Programs Summary, Vol. VI, consists of: an unclassified digest of appropriate material from Vols. I, II, and III; an original presentation of the JPL quality assurance and reliability efforts, and the environmental- and dynamic-testing facility-development activities; and a reprint of the space science instrumentation studies of Vols. I and II. This instrumentation work is conducted by the JPL Space Sciences Division and also by individuals of various colleges, universities, and other organizations. All such projects are supported by the Laboratory and are concerned with the development of instruments for use in the NASA space flight programs.

Approved by:

W. H. Pickering

W. H. Pickering, *Director*

Jet Propulsion Laboratory

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I. *Surveyor* Project

THE LUNAR PROGRAM

A. Introduction

The *Surveyor* flight spacecraft are designed to span the gap between the *Ranger* Project and the *Apollo* Project by making soft landings on the moon to extend our knowledge of lunar conditions and determine the suitability of sites for proposed *Apollo* spacecraft landings.

Hughes Aircraft Company (HAC), Space Systems Division, is under contract to fabricate the *Surveyor* A-21 spacecraft. The launch vehicle, a combination *Atlas/Centaur*, is provided by General Dynamics/Convair. Control, command, and tracking functions for the *Surveyor* missions are performed by the *Surveyor* Mission Operations System and the JPL Deep Space Network.

Surveyor I, the first flight spacecraft, was launched from Cape Kennedy, Florida, on May 30, 1966, and soft-landed on the moon on June 2, 1966. By June 14, when lunar sunset occurred, approximately 100,000 commands had been received by the spacecraft, and 10,338 pictures of the spacecraft and its immediate vicinity had been transmitted. During the second lunar day, 812 additional pictures were transmitted by *Surveyor I*.

Surveyor II, the second flight spacecraft, was launched from Cape Kennedy on September 20, 1966. The spacecraft performed nominally until the command was given for midcourse thrust execution. At that time, one vernier engine did not ignite. The resulting imbalance of thrust from the other two vernier engines imposed a tumbling motion on the spacecraft, from which it failed to recover. The *Surveyor II* mission was terminated on September 22 after all contact with the spacecraft had been lost.

Surveyor III, the third flight spacecraft, was launched from Cape Kennedy, Florida, on April 17, 1967, and soft-landed on the moon, in a crater of the Ocean of Storms, on April 19, 1967. By the end of the first lunar day, the spacecraft had responded to more than 30,000 commands and had transmitted 6315 pictures of the spacecraft and its immediate vicinity. The first photographs of an eclipse of the sun by the earth were taken by *Surveyor III*. Successful surface sampler experiments were conducted by the spacecraft; four trenches were dug in the lunar soil, the largest of which was 2 in. wide, 10 in. long, and 7½ in. deep. Then operations were discontinued until the dawn of the second lunar day. (Details of the *Surveyor III* launch and mission operations will be covered in subsequent SPS issues.)

B. Payload and Scientific Mechanisms

1. Alpha-Scattering Experiment

The alpha-scattering experiment on spacecraft SC-5 through -7 will be used for qualitative and quantitative composition analysis of lunar surface materials. The lunar surface will be bombarded with 6-Mev alpha particles from sources contained in the instrument. Backward scattered alpha particles, as well as protons generated within the lunar surface materials by the incident alpha particles, will be detected by solid-state detectors. The instrument is capable of detecting the presence of all elements except hydrogen and helium with a detection threshold of approximately 1% by weight.

The alpha-scattering instrument consists of three parts—a sensor head, electronics, and a standard sample assembly. The required electrical interfaces between the instrument and the spacecraft are provided by an auxiliary unit which affords command decoding, signal processing, and power management, and by direct connection to the basic bus, which provides instrument clocks, temperature sensor current, and commutation of engineering data. The electronics and auxiliary units are contained in a new thermal compartment C. A heater maintains the compartment C radiator within the operating temperature range of the auxiliary and electronic units.

A deployment mechanism (Fig. 1) attached to the spacecraft provides for stowage and deployment of the sensor head. The experiment is operated in three positions: (1) stowed position, utilizing a standard sample contained in the deployment mechanism, (2) background count position, approximately 15 in. from the lunar surface, and (3) on the lunar surface.

During this reporting period, the alpha-scattering experiment was successfully integrated into the SC-5 spacecraft system, and all experiment data channels were calibrated. The alpha-scattering subsystem was operated during the SC-5 mission sequence/electromagnetic interference test. The testing included prelaunch checkout (systems readiness test) and coast-phase and post-touchdown operations. RF interference occurred during prelaunch systems readiness testing, but subsequent special tests indicated that the interference would not prevent verification of the alpha-scattering experiment readiness condition in the launch pad environment. Integration of the SC-6 alpha-scattering subsystem with the SC-6 spacecraft is currently under way.

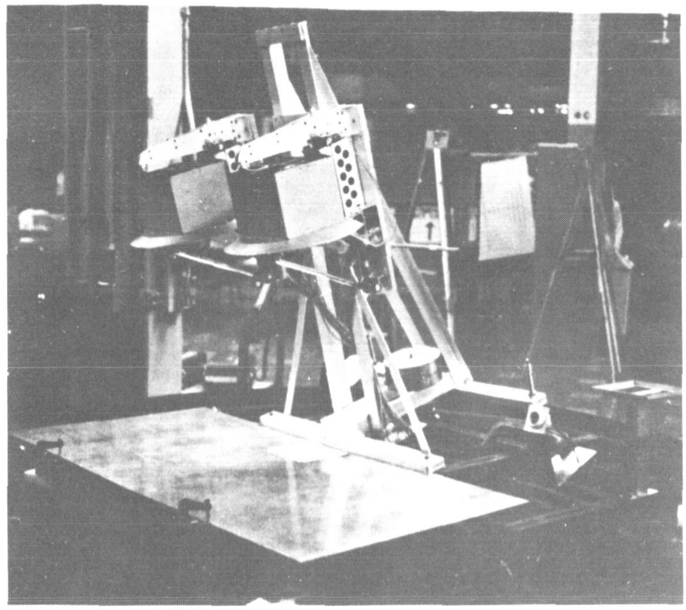


Fig. 1. Alpha-scattering experiment deployment mechanism

2. Surface Sampler

The surface sampler subsystem is part of the SC-3 and -4 payloads. It is designed to manipulate a portion of the lunar surface near the spacecraft. Survey TV camera 3 permits viewing of these operations. The visual data obtained during such operations as picking, digging, scraping, and trenching, combined with an assessment, through motor current telemetry, of the forces being developed during these operations, should provide significant information related to lunar surface characteristics.

The surface sampler subsystem consists of a mechanism, an auxiliary electronics and thermal compartment, and installation substructure and wiring harnesses. The mechanism performing operations on the lunar surface is controlled by the auxiliary electronics housed in its own thermal compartment. The auxiliary electronics receives and decodes ground commands, and provides power and signal conditioning for the surface sampler subsystem.

During this reporting period, surface sampler 1 successfully completed a special motor current/force calibration and TV viewability test with the SC-3 spacecraft at the Eastern Test Range. Surface sampler 2 successfully completed a series of system level tests with the SC-4 spacecraft.

II. *Mariner* Venus 67 Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

The primary objective of the *Mariner* Venus 67 Project is to conduct a flyby mission to Venus in 1967 to obtain scientific information which will complement and extend the results obtained by *Mariner II* relevant to determining the origin and nature of Venus and its environment. Secondary objectives are to: (1) acquire engineering experience in the conversion of a spacecraft designed for a mission to Mars (spare flight spacecraft from *Mariner* Mars 1964 Project) into one designed for a mission to Venus and in the operation of such a spacecraft, and (2) obtain information on the interplanetary environment during a period of increasing solar activity. An *Atlas/Agena D* launch vehicle will be used.

Due to the minimum length of time (18 months) between authorization of the project and the launch opportunity, techniques and hardware developed during prior projects must be utilized to the fullest extent possible. The single flight spacecraft, designated M67-2, will be a converted *Mariner* Mars 1964 Project flight spare. Portions of the proof test model and certain critical spare units from the *Mariner* Mars 1964 Project are being prepared for use as a flight support spacecraft, designated M67-1. The flight support spacecraft will serve the double function of a pseudo-proof test model and a backup spacecraft for qualifying spare subsystems.

Various changes to the *Mariner* Mars 1964 spacecraft design are necessitated by the fact that the M67-2 flight spacecraft will travel toward, rather than away from, the sun; also, conversions must be made to accommodate the revised encounter sequencing and science payload.

The S-band radio occultation experiment, one of seven scientific experiments approved for the mission, requires the use of only the RF transmission subsystem on the spacecraft. The celestial mechanics experiment uses only the tracking doppler data derived from the RF carrier. The ultraviolet photometer, helium magnetometer, solar plasma probe, and trapped radiation detector experiments are to be accomplished using existing instrumentation with only minor modifications. Only the dual-frequency radio propagation experiment requires the incorporation of a new scientific instrument into the payload. The Principal Scientific Investigators for these experiments are given in Table 1.

Table 1. *Mariner* Venus 67 Principal Scientific Investigators

Experiment	Principal Scientific Investigator	Affiliation
S-band radio occultation	A. J. Kliore	Jet Propulsion Laboratory
Ultraviolet photometer	C. A. Barth	University of Colorado
Dual-frequency radio propagation	V. R. Eshleman	Stanford University
Helium magnetometer	E. J. Smith	Jet Propulsion Laboratory
Solar plasma probe	H. S. Bridge	Massachusetts Institute of Technology
Trapped radiation detector	J. A. Van Allen	State University of Iowa
Celestial mechanics	J. D. Anderson	Jet Propulsion Laboratory

During this reporting period, the M67-2 spacecraft, with a full complement of flight hardware, successfully completed its second system test. Both the M67-1 and M67-2 spacecraft were given preshipment tests and were then shipped to the Eastern Test Range. At the Eastern Test Range, *Atlas/Agena D* launch vehicle preparations are proceeding normally. Detailed integration and conversion schedules were established for Launch Complexes 12 and 13.

B. Design and Development

1. Spacecraft Mechanical Configuration

The *Mariner* Mars 1964 basic bus structure and packaging arrangement were utilized in the *Mariner* Venus 67 spacecraft design. Modifications of this basic structure were limited to add-on bracketry to support relocated equipment. Rearrangement of much of the equipment was necessary because of characteristics unique to the Venus trajectory and revisions in the science payload (four experiments were dropped and one experiment was added). The final version of the *Mariner* Venus 67 spacecraft configuration through the detail design period is illustrated in Fig. 1.

a. Antennas. To satisfy earth-pointing requirements, the omnidirectional and high-gain antennas must be oriented in the anti-sun direction. The size of these antennas dictated that the under-shroud orientation could not differ greatly from that of the *Mariner* Mars 1964 spacecraft without imposing serious design changes. These restrictions made it necessary to locate the antennas in the same positions as on the *Mariner* Mars 1964 spacecraft. Thus, the in-flight attitude of the *Mariner* Venus 67 spacecraft had to be reversed as compared to that of the *Mariner* Mars 1964 spacecraft. Specifically, the *Mariner* Mars 1964 low-gain antenna and all support bracketry were utilized in the *Mariner* Venus 67 spacecraft design. However, the high-gain antenna support structure was modified to provide proper pointing direction and mechanization to change the pointing direction by discrete steps for increased coverage during occultation and after Venus encounter.

b. Solar panels. The *Mariner* Venus 67 solar panels are the same length as those used on the *Mariner* Mars 1964 spacecraft, but the cell area is 40 ft^2, instead of 70 ft^2. Because of the reversed sun-line attitude required for the *Mariner* Venus 67 spacecraft, the solar cell area was located away from the bus. Solar panel deployment springs and cruise dampers were unchanged. The boost latching and dampers were replaced with orthogonal pairs of dampers that tie the panels together at the tips. Four pinpullers, instead of the eight required for *Mariner* Mars 1964, are used to release the panels. Solar-panel-mounted equipment includes the dual-frequency radio antennas and the temperature control references. Orientation and phasing of the dual-frequency radio antennas provide peak gain in the direction of earth at encounter.

c. Attitude-control sensors. For the *Mariner* Venus 67 spacecraft, the primary and secondary sun sensors were interchanged because of the sun attitude required. To provide an improved field of view, the Canopus sensor was relocated from the bottom ring of Bay VIII to a centerline location on the top ring of the same bay. A modified *Mariner* Mars 1964 earth sensor and a terminator sensor were adapted, with suitable bracketry, to mount on the Bay VI and Bay II secondary sun sensor pedestals, respectively. A planet sensor was located on the upper ring in Bay VIII with indexing surfaces to align the instrument centerline to an angle approximately 25 deg from the sun line.

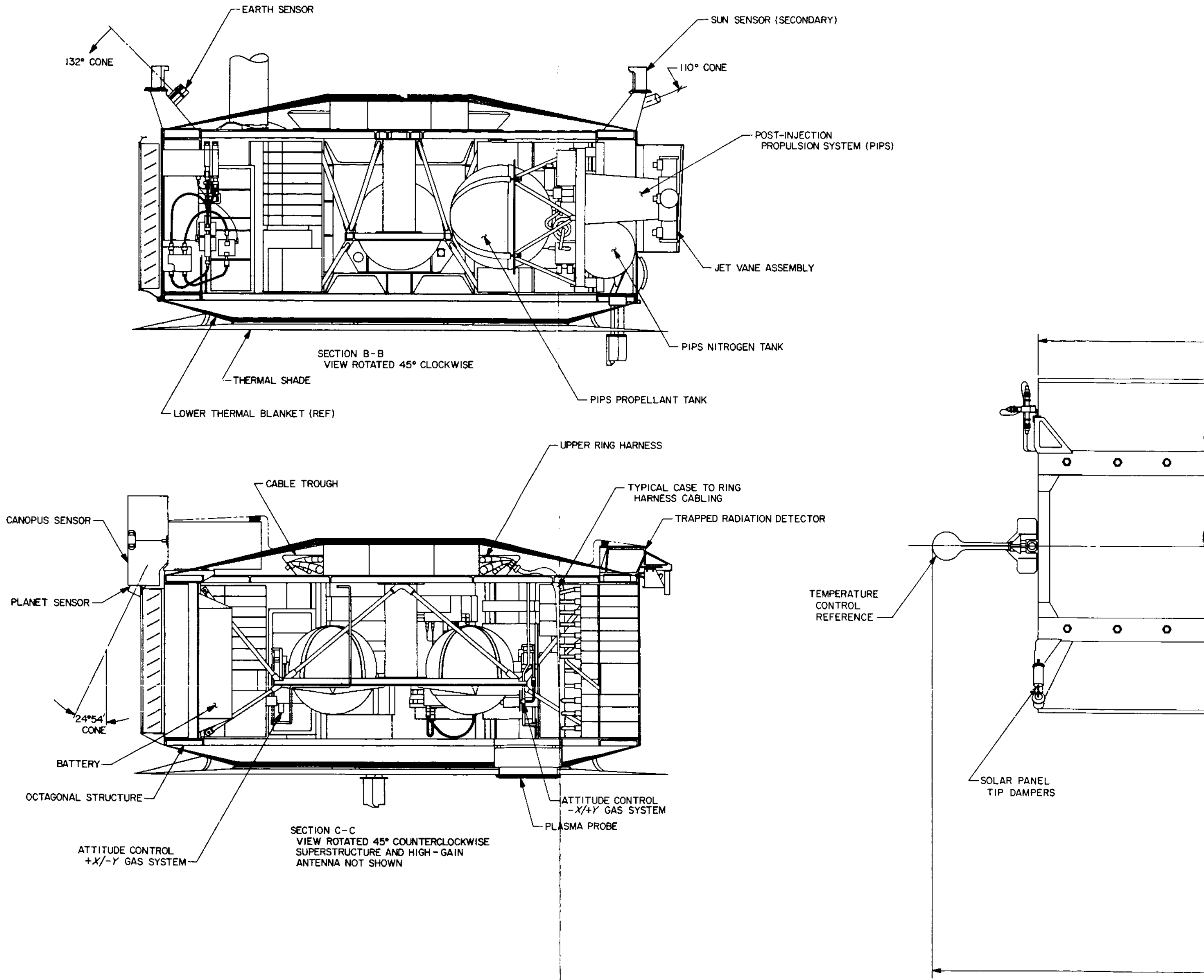
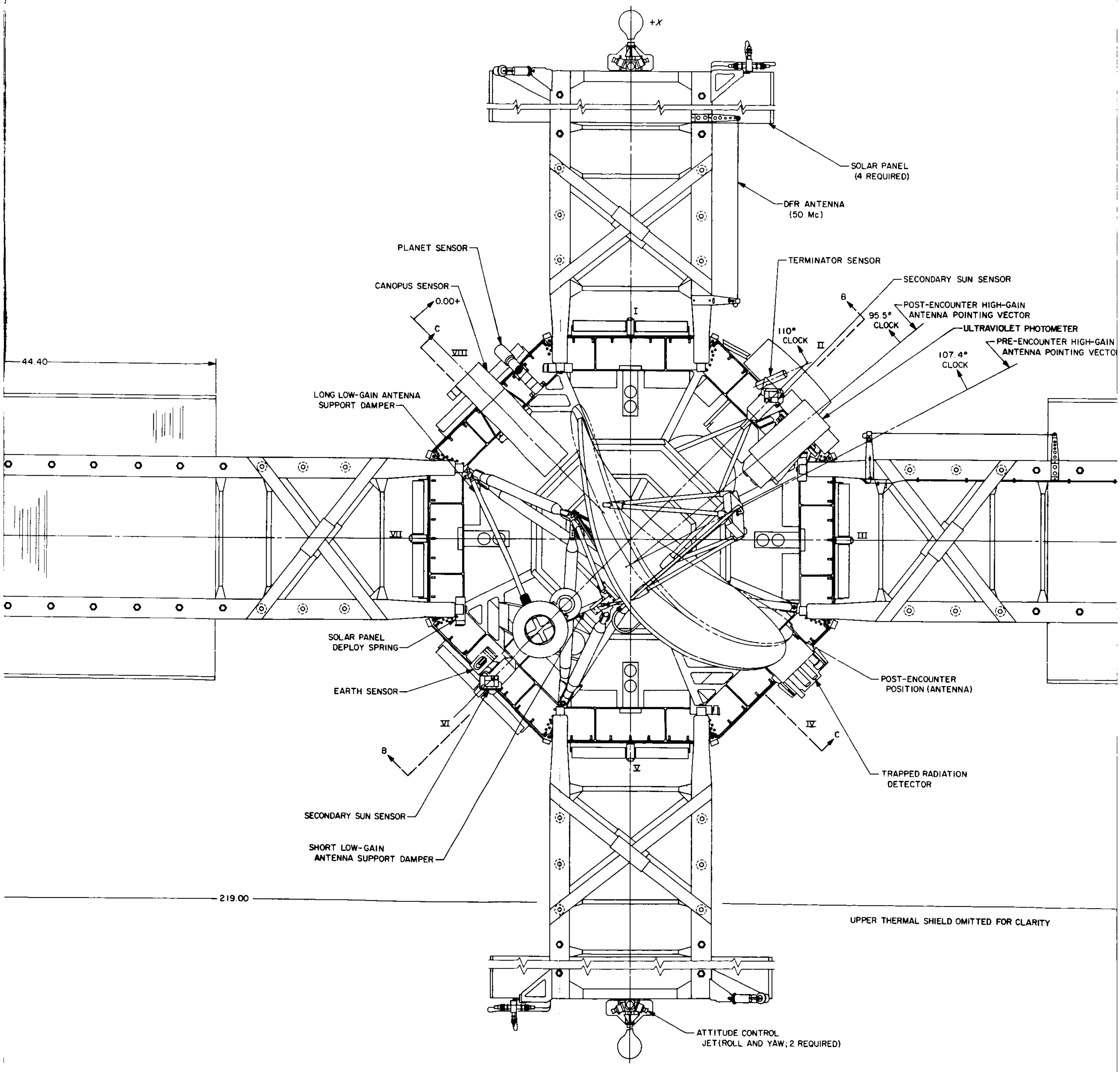
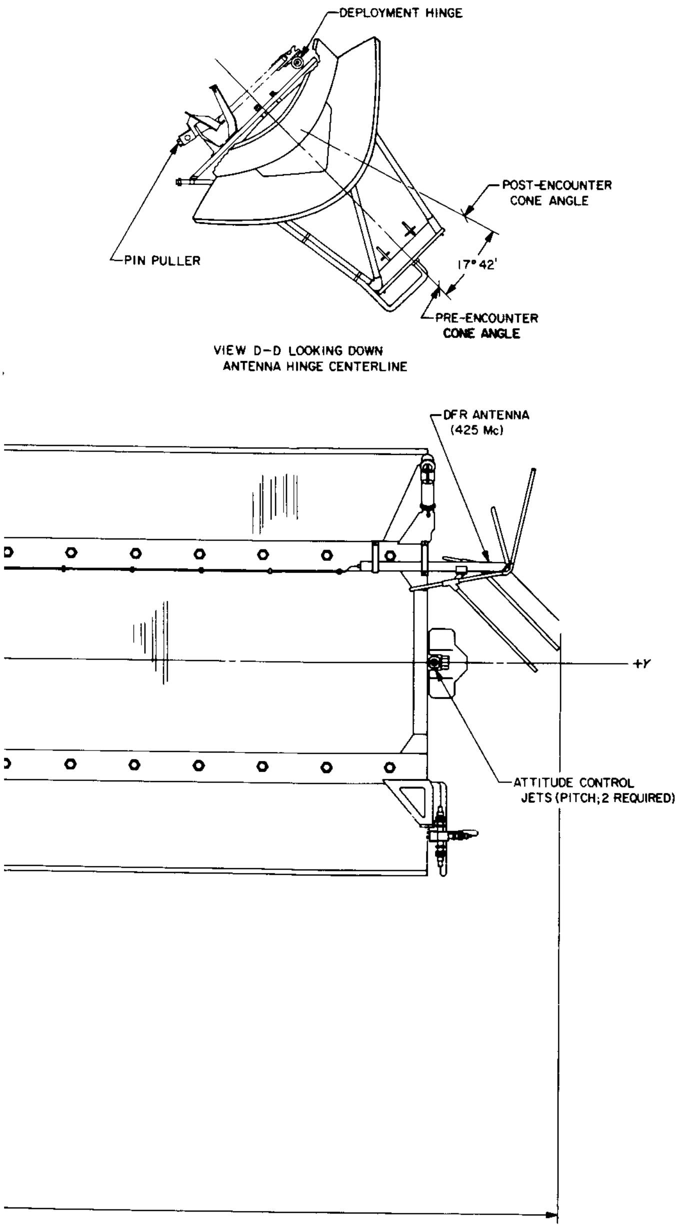


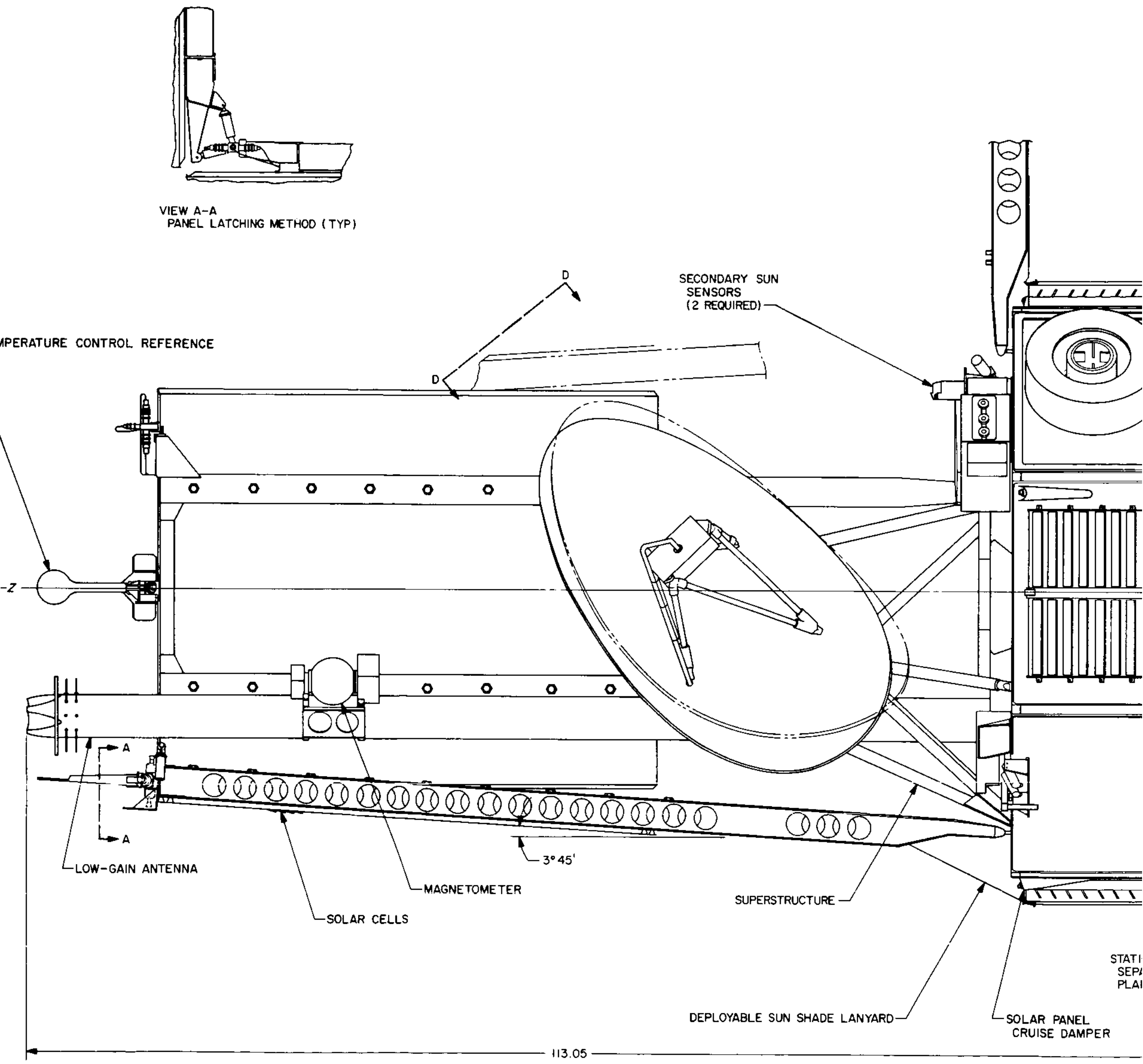
Fig. 1. ***Mariner*** **Venus 67 spacecraft**



+X
SOLAR PANEL (4 REQUIRED)
DFR ANTENNA (50 Mc)
PLANET SENSOR
CANOPUS SENSOR
0.00+
TERMINATOR SENSOR
SECONDARY SUN SENSOR
POST-ENCOUNTER HIGH-GAIN ANTENNA POINTING VECTOR
95.5° CLOCK
110° CLOCK
ULTRAVIOLET PHOTOMETER
PRE-ENCOUNTER HIGH-GAIN ANTENNA POINTING VECTO
107.4° CLOCK
44.40
LONG LOW-GAIN ANTENNA SUPPORT DAMPER
SOLAR PANEL DEPLOY SPRING
EARTH SENSOR
POST-ENCOUNTER POSITION (ANTENNA)
TRAPPED RADIATION DETECTOR
SECONDARY SUN SENSOR
SHORT LOW-GAIN ANTENNA SUPPORT DAMPER
219.00
UPPER THERMAL SHIELD OMITTED FOR CLARITY
ATTITUDE CONTROL JET(ROLL AND YAW; 2 REQUIRED)


DEPLOYMENT HINGE
POST-ENCOUNTER CONE ANGLE
PIN PULLER
17° 42'
PRE-ENCOUNTER CONE ANGLE
VIEW D–D LOOKING DOWN ANTENNA HINGE CENTERLINE
DFR ANTENNA (425 Mc)
+Y
ATTITUDE CONTROL JETS (PITCH; 2 REQUIRED)

6-2


VIEW A-A
PANEL LATCHING METHOD (TYP)
D
D
SECONDARY SUN
SENSORS
(2 REQUIRED)
TEMPERATURE CONTROL REFERENCE
-Z
A
A
LOW-GAIN ANTENNA
3° 45'
MAGNETOMETER
SOLAR CELLS
SUPERSTRUCTURE
DEPLOYABLE SUN SHADE LANYARD
SOLAR PANEL
CRUISE DAMPER
113.05

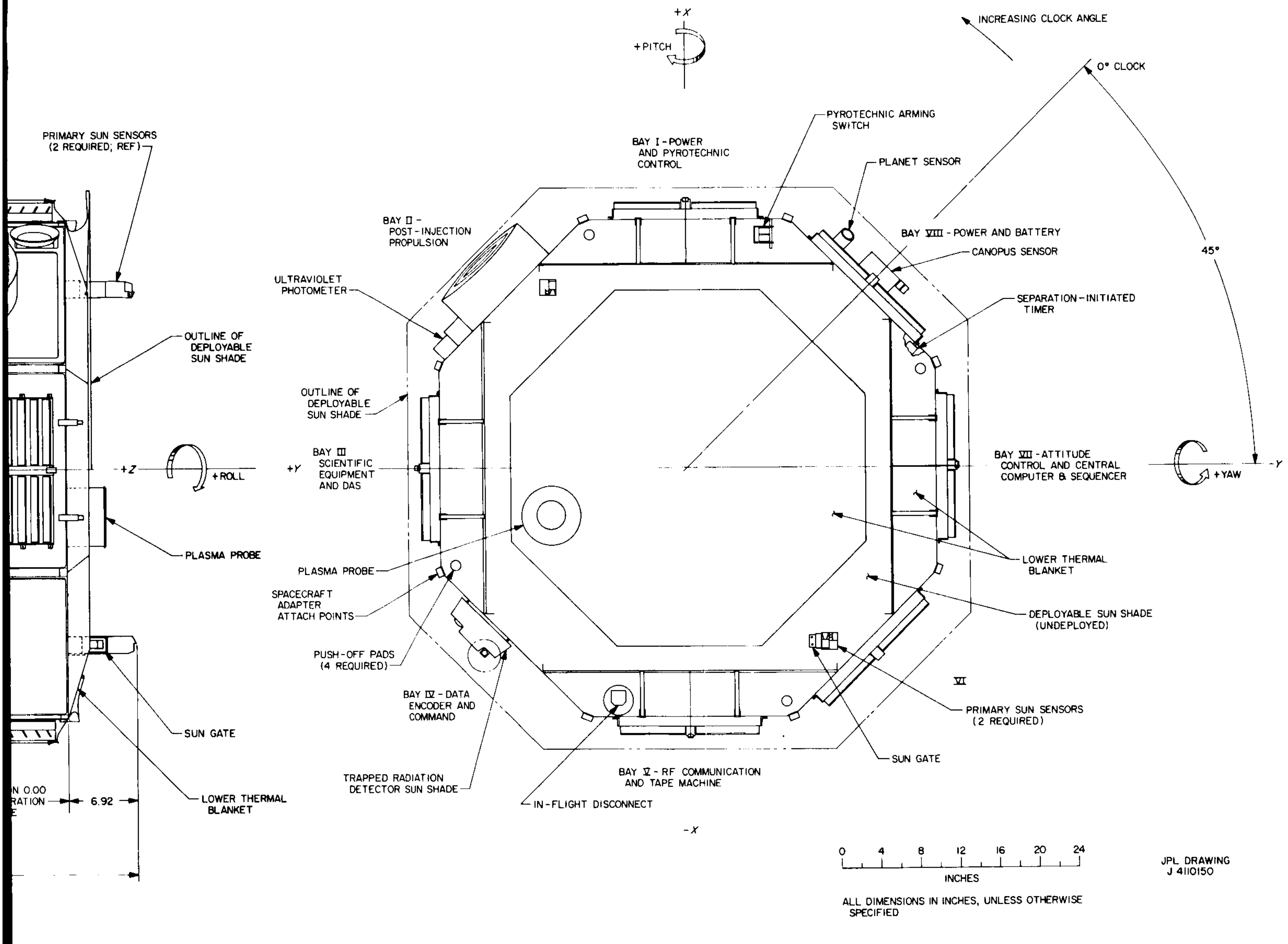

+X
+PITCH
INCREASING CLOCK ANGLE
0° CLOCK
45°
PRIMARY SUN SENSORS (2 REQUIRED; REF)
PYROTECHNIC ARMING SWITCH
BAY I - POWER AND PYROTECHNIC CONTROL
PLANET SENSOR
BAY II - POST-INJECTION PROPULSION
BAY VIII - POWER AND BATTERY
CANOPUS SENSOR
ULTRAVIOLET PHOTOMETER
SEPARATION-INITIATED TIMER
OUTLINE OF DEPLOYABLE SUN SHADE
OUTLINE OF DEPLOYABLE SUN SHADE
BAY III SCIENTIFIC EQUIPMENT AND DAS
-Z
+ROLL
+Y
BAY VII - ATTITUDE CONTROL AND CENTRAL COMPUTER & SEQUENCER
+YAW
-Y
PLASMA PROBE
PLASMA PROBE
LOWER THERMAL BLANKET
SPACECRAFT ADAPTER ATTACH POINTS
DEPLOYABLE SUN SHADE (UNDEPLOYED)
PUSH-OFF PADS (4 REQUIRED)
VI
BAY IV - DATA ENCODER AND COMMAND
PRIMARY SUN SENSORS (2 REQUIRED)
SUN GATE
SUN GATE
BAY V - RF COMMUNICATION AND TAPE MACHINE
TRAPPED RADIATION DETECTOR SUN SHADE
IN-FLIGHT DISCONNECT
LOWER THERMAL BLANKET
6.92
-X
0 4 8 12 16 20 24
INCHES
ALL DIMENSIONS IN INCHES, UNLESS OTHERWISE SPECIFIED
JPL DRAWING J 4110150

Fig. 1. (contd)

d. Science instruments. The trapped radiation detector was mounted at the same location as that of the *Mariner* Mars 1964, with the instrument reversed to provide proper sun-line orientation. The magnetometer and ion chamber experiments remained in the *Mariner* Mars 1964 location on the low-gain antenna, but some thermal control effort was required to ensure acceptable temperatures. The required field of view for the cosmic ray telescope was provided by reversing the instrument module in the science chassis and by making a reinforced cutout in the spacecraft structure. The plasma probe was moved from the solar panel and placed on an auxiliary support structure inboard of the lower ring on the sun side of the spacecraft. The ultraviolet photometer was located over Bay II at a cone angle of 90 deg, to allow a field of view compatible with the planet encounter geometry. The dual-frequency radio receiver, the new experiment that was added to the science payload, was packaged as an electronics subassembly and placed in the Bay III electronics assembly.

e. Exterior thermal hardware. Shield design and temperature control louver hardware remained the same as used in the *Mariner* Mars 1964 spacecraft, with the exception of the sun-side thermal blanket. In order to prevent heat leaks at Venus intensities, a shield was designed for sun-side mounted equipment. The mechanization consisted of a deployable shade that is erected following launch vehicle separation. All hardware interfacing with the launch vehicle adapter will be shielded from the solar input by this sun shade.

2. Canopus Sensor

The *Mariner* Venus 67 Canopus sensor is a modification of the *Mariner* Mars 1964 Canopus sensor. Rapid Canopus acquisition has been sacrificed in favor of mechanization simplicity. However, the major change involves reducing the straylight sensitivity by redesigning the shade, optics, and sun shutter.

The requirement to track Canopus in the proximity of Venus causes the straylight problem. The illumination on the Canopus sensor due to Venus exceeds 2 billion times the illumination due to Canopus. Venus light coming from outside the 5 $\times$ 32-deg field of view of the sensor is scattered off the various optical components and onto the image dissector photocathode area defining the field of view.

A two-stage shade (Fig. 2) was designed to improve the straylight sensitivity of the *Mariner* Venus 67 Canopus sensor. The first-stage shade prevents light from outside a critical field of view from falling directly on the sensor lens; the second-stage shade prevents light emanating from outside a second critical field of view from falling directly onto the first-stage shade defining aperture. The second-stage shade defining aperture is located so that it is not visible from any point on the sensor lens.

A four-element lens system was designed (Fig. 3). In general, an attempt was made to use oversized elements to minimize edge reflections as a source of straylight. The edges of the elements are polished and painted with cat-a-lac black paint to minimize stray reflections. The unused area on the last surface is also painted black.

An internal sun shutter was designed, using a rotary solenoid. The solenoid utilizes a balanced helical magnetic structure to generate a stroke of 36 deg. Armco magnetic iron was selected for the solenoid because it has low retentivity. Low retentivity was important for two reasons: general spacecraft magnetic cleanliness, and Canopus sensor null accuracy. The solenoid was located 3 in. from the image dissector to limit null offsets to less than 0.05 deg.

3. Battery

The basic design of the battery is dictated by the system energy requirements at various critical phases of the mission. The *Mariner* Venus 67 power requirements during launch and encounter are 1200 and 900 w-hr, respectively. These power requirements are the same as those that were required for the *Mariner* Mars 1964 mission. Since the *Mariner* Mars 1964 battery performed well during the mission, the same basic design is being used for the *Mariner* Venus 67 batteries, with modifications to improve reliability, lifetime, and performance.

The vent tube hole of the potting cover was removed to eliminate the weak spot in the cell design and to allow a continuous integral seal around the entire edge of the cover. The container was molded in Cycolac T-2502 and the cover plug in polystyrene RMD-4511 to provide a cell structure of high strength. On the negative plate, the active material, polyvinyl alcohol binder, was replaced with Teflon powder. The active material mix is now 90% ZnO, 7% HgO, and 3% Teflon powder.

Testing results and performance evaluation indicate that the *Mariner* Venus 67 batteries surpass the *Mariner* Mars 1964 batteries in performance at environmental

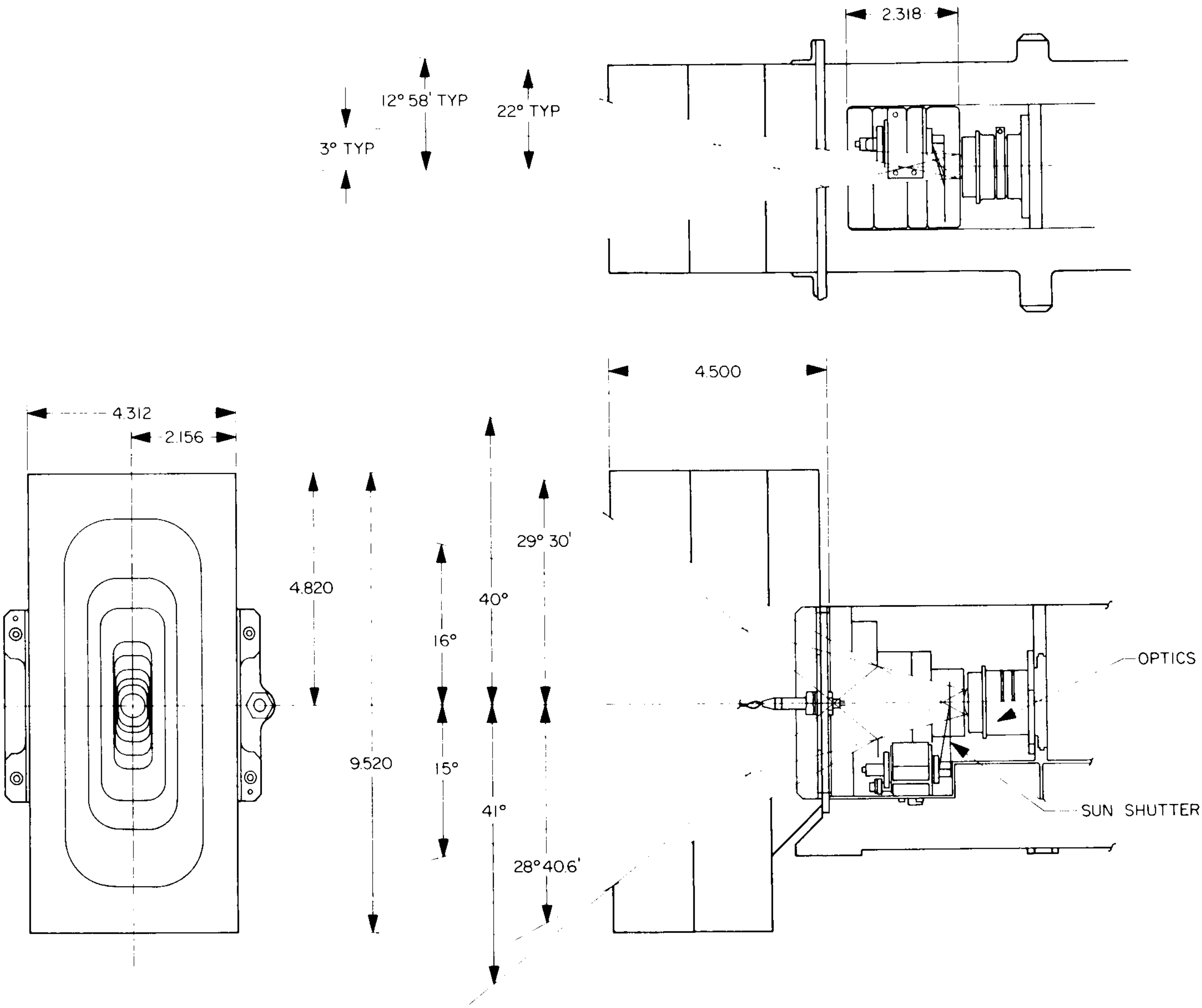


Fig. 2. Canopus sensor two-stage shade

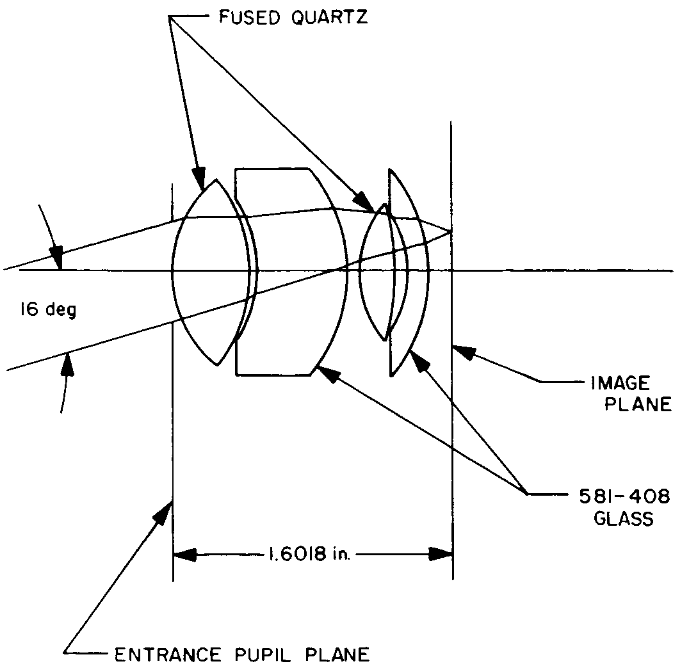


Fig. 3. Canopus sensor lens

extremes. These performance increases more than compensate for the slight additional weight of the *Mariner* Venus 67 battery.

C. Testing Equipment and Operations

1. Space Simulator Tests

The M67-2 spacecraft underwent system and thermal testing in the JPL 10-ft space simulator. The tests were similar to those performed with the M67-1 spacecraft (SPS 37-44, Vol. VI, p. 7). Results of this test, together with results of the corresponding M67-1 test, provide the following conclusions:

(1) The thermal design is adequate for all anticipated combinations of spacecraft operating mode and flight environment.

(2) All flight telemetry temperatures are within design accuracy.

(3) Operational support equipment installed in the simulator during the system portion of the test had only a minor influence on bus temperature.

(4) Discrepancies of 6 to 12% still exist between the various intensity monitors used, although this disagreement is less for M67-2 than that observed during the M67-1 test.

(5) M67-2 has thermal characteristics similar to M67-1.

2. Power Subsystem

The flight and flight spare units of the power subsystem underwent system testing on the spacecraft. Results of the tests were very satisfactory. The power subsystem operated properly, except for several minor problems that are being resolved. Special tests using the science-exclusive circuit and the boost mode of the charger were run successfully several times. The exclusive circuit turns off science if the main booster regulator fails (the maneuver booster takes over the load) and the gyros are on. This circuit may be overridden with the command which turns science on. This commanded override was also tested successfully.

III. *Mariner* Mars 1969 Project

THE PLANETARY–INTERPLANETARY PROGRAM

A. Introduction

The primary objective of the *Mariner* Mars 1969 Project is to conduct two flyby missions to Mars in 1969 to make exploratory investigations of the planet which will set the basis for future experiments — particularly those relevant to the search for extraterrestrial life. The secondary objective is to develop the technology needed for succeeding Mars missions.

The spacecraft design concept will be based on that of the successful *Mariner IV* spacecraft developed under the *Mariner* Mars 1964 Project. However, considerable modifications will be made to meet the 1969 mission requirements and to enhance mission reliability.

The launch vehicle will be the *Atlas/Centaur SLV-3C*. This vehicle, developed under contract for and direction of the Lewis Research Center by General Dynamics/Convair, has a single- or double-burn capability in its second stage and a considerably increased performance rating over the *Atlas D/Agena D* used in the *Mariner IV* mission.

Mariner Mars 1969 missions will be supported by the Eastern Test Range launch facilities at Cape Kennedy, the tracking and data acquisition facilities of the Deep Space Network, and other NASA facilities.

The six planetary-science experiments selected by NASA for the *Mariner* Mars 1969 missions are the following: TV, infrared spectrometer, ultraviolet airglow spectrometer, infrared radiometer, S-band occultation, and celestial mechanics. Additionally, a planetary-approach-guidance engineering experiment will be incorporated to test the feasibility and flight performance of onboard optical sensors and associated data processing techniques necessary for optical approach guidance. The Scientific Investigators for the planetary-science experiments are listed in Table 1.

During this reporting period, a series of subsystem preliminary design reviews was completed. Spacecraft functional requirements are being revised to reflect design changes negotiated as a result of the subsystem reviews, and the previous system review. The operational support equipment design is nearing completion.

Table 1. *Mariner* Mars 1969 Scientific Investigators

Experiment	Scientific Investigator	Affiliation
TV	R. B. Leighton[a]	California Institute of Technology
	B. C. Murray	California Institute of Technology
	R. P. Sharp	California Institute of Technology
	N. H. Horowitz	California Institute of Technology
	J. D. Allen	Jet Propulsion Laboratory
	A. G. Herriman	Jet Propulsion Laboratory
	L. R. Malling	Massachusetts Institute of Technology
	R. K. Sloan	Jet Propulsion Laboratory
	N. Davies	Rand Corporation
	C. Leovy	Rand Corporation
Infrared spectrometer	G. C. Pimentel[a]	University of California, Berkeley
	K. C. Herr	University of California, Berkeley
Ultraviolet airglow spectrometer	C. A. Barth[a]	University of Colorado
	W. G. Fastie	Johns Hopkins University
Infrared radiometer	G. Neugebauer[a]	California Institute of Technology
	G. Munch	California Institute of Technology
	S. C. Chase	Santa Barbara Research Center
S-band occultation	A. J. Kliore[a]	Jet Propulsion Laboratory
	D. L. Cain	Jet Propulsion Laboratory
	G. S. Levy	Jet Propulsion Laboratory
Celestial mechanics	J. D. Anderson[a]	Jet Propulsion Laboratory

[a]Principal Scientific Investigator.

The mission operations design effort is progressing. Most of the functional requirements have been completed and approved, and a majority have been issued. The software design is also under way.

B. Design and Development

1. Spacecraft Subsystems

As a result of the system design review in November 1966, and the subsystem design reviews which were recently completed, changes were made at the subsystem level to the *Mariner* Mars 1969 spacecraft design. The principal design changes are given in Table 2. A comparison of *Mariner* Mars 1964 and *Mariner* Mars 1969 subsystem weights, and total spacecraft weights, is given in Table 3.

2. Science Data Handling

The science data formats which will be used for each of the possible modes of encounter and playback are the real-time science format, the digital recorder format, and the analog recorder format. The real-time science format is used only in nominal encounter. It consists of 28 10-bit words conveying calibration and diagnostic engineering telemetry on the instruments, some science from the infrared and ultraviolet spectrometers, and all data from the infrared radiometer. The digital recording format includes the infrared and ultraviolet spectrometer data, TV coding, the real-time science data, and some TV picture data (every seventh picture element). The center section of each TV line is eliminated to provide room for the non-TV data in this format. The analog recording format contains TV analog data (934 picture elements per line), data automation system TV coding, and a digitized TV selection consisting of 4 bits of every 28th element and all 8 bits of 20 selected elements from each line.

During nominal encounter, the data returned to the Deep Space Instrumentation Facility (DSIF) will be in the real-time science format and will be returned at 66⅔ bits/sec. During nominal playback, the entire content of both the digital and the analog tape recorders will be sent to the DSIF at 270 bits/sec. During optional encounter, the entire format of the digital recorder will be received at the DSIF at 16.2 kbits/sec. This can only be decommutated at the Mars Deep Space Station at Goldstone, California. During optional playback, the entire format of the analog recorder will be received at the DSIF at 16.2 kbits/sec. This, too, can only be decommutated at the Mars Deep Space Station at Goldstone. The flight data flow during the nominal encounter mode is shown in Fig. 1.

Table 2. *Mariner* Mars 1969 spacecraft design changes

Subsystem components	Former design	Present design
Radio frequency high-gain antenna	*Mariner* Mars 1964	40-in. circular aperture
Traveling wave tube amplifier	10-w tube (unspecified manufacturer)	10/20-w *Apollo* tube
Flight command detector	*Mariner* Mars 1964	Automatic acquisition detector, with 1.5-db improvement in system performance
2400-Hz power inverters	One main 2400-Hz inverter and one maneuver 2400-Hz inverter; single clock, integral with main inverter	Two redundant 2400-Hz inverters (with integral clocks)
Central computer & sequencer (CC&S) scan pointing control	No scan pointing control except via data automation system	CC&S performs both cone and clock slews directly, the clock slew being programmable in flight
Flight telemetry synchronization	PN synchronization carried on separate channel	No sync channel
Data channels	One data channel: engineering data during cruise; engineering and real-time science time shared during encounter; and all digital tape recorder data (2 hr of cruise/day during playback)	Three possible data channels, with a maximum of two being used at any one time: (1) engineering channel; (2) science channel; (3) block coded channel
Subframe sync	One 7-bit, all ones word per subframe	15-bit PN sequence starts out each subframe
Timing reference	307.2 kHz and 2400 Hz from power	480.0-kHz crystal oscillator
Engineering rates	8⅓, 33⅓, and 66⅔ bits/sec	8⅓ and 33⅓ bits/sec
Subcommutation identification	Medium-rate word identifying positions of low and low-low decks	High-rate word identifying positions of medium, low, and low-low decks
Ground telemetry subsystem (GTS)	GTS demodulator prime; GTS decommutator back-up to telemetry and command processor	None; all demodulation and decommutation performed by telemetry and command processor
Attitude control gyro turn-off	Ground command for gyro turn-off, which also removed Canopus brightness gate	Independent ground command for gyro turn-off
Propulsion catalyst	Oxidizer required to raise catalyst to proper temperature	Spontaneous catalyst; requires no oxidizer
Temperature control active control	Resistive heating only	Resistive heating and promethium heat source
Data storage playback rates	66⅔ and 33⅓ bits/sec	270 bits/sec only
Digital recorder record rates	16.2 and 8.1 kbits/sec	16.2 kbits/sec only
Digital data playback	Single playback of digital data	Double playback of digital data before first transfer
Analog data playback	Playback via digital tape recorder only	Playback via digital tape recorder at 270 bits/sec or playback via block coder in flight telemetry subsystem at 16.2 kbits/sec
Power	400 Hz for motors; 2400 Hz for electronics; 2400 Hz on continuously	2400 Hz only; switched off during cruise
Data automation encounter timer	Timed digital-tape-recorder speed change, platform slews, and analog-tape-recorder and digital-tape-recorder shutoff	Deleted; functions either assumed by CC&S or no longer required
16.2-kbits/sec encounter science data	Recorded on digital tape recorder only	Recorded on digital tape recorder and sent to flight telemetry subsystem for possible block coded transmission in real time
66⅔-bits/sec real-time science	Time-shared channel with engineering data	Transmitted continuously during encounter
Scan near-encounter clock reference	Two references; one for initial position and one for slewed position	One reference only; programmed in real time for slew
TV covers	No covers	Two transparent, optically flat covers

Table 3. Spacecraft weight summary

Subsystem	*Mariner* Mars 1964, lb	*Mariner* Mars 1969, lb
Structure	74.23	170.4
Radio frequency	44.51	58.2
Flight command	10.16	11.4
Power	150.75	106.5
CC&S	11.75	24.0
Flight telemetry	21.97	22.4
Attitude control	62.93	59.6
Pyrotechnic	12.64	12.0
Cabling	45.75	68.7
Propulsion	47.42	46.4
Temperature control	17.74	38.3
Mechanical devices	—	43.1
Approach guidance	—	10.5
Data storage	17.11	38.0
Data automation	11.12	19.5
Cruise science	—	—
Scan control	6.73	18.3
UV photometer	6.46	—
UV spectrometer	—	30.6
Television	11.17	47.6
IR spectrometer	—	29.0
IR radiometer	—	5.0
Total spacecraft design		
Estimate	575.61	859.5
Engineering contingency	—	40.0
Total spacecraft weight	575.61	899.5

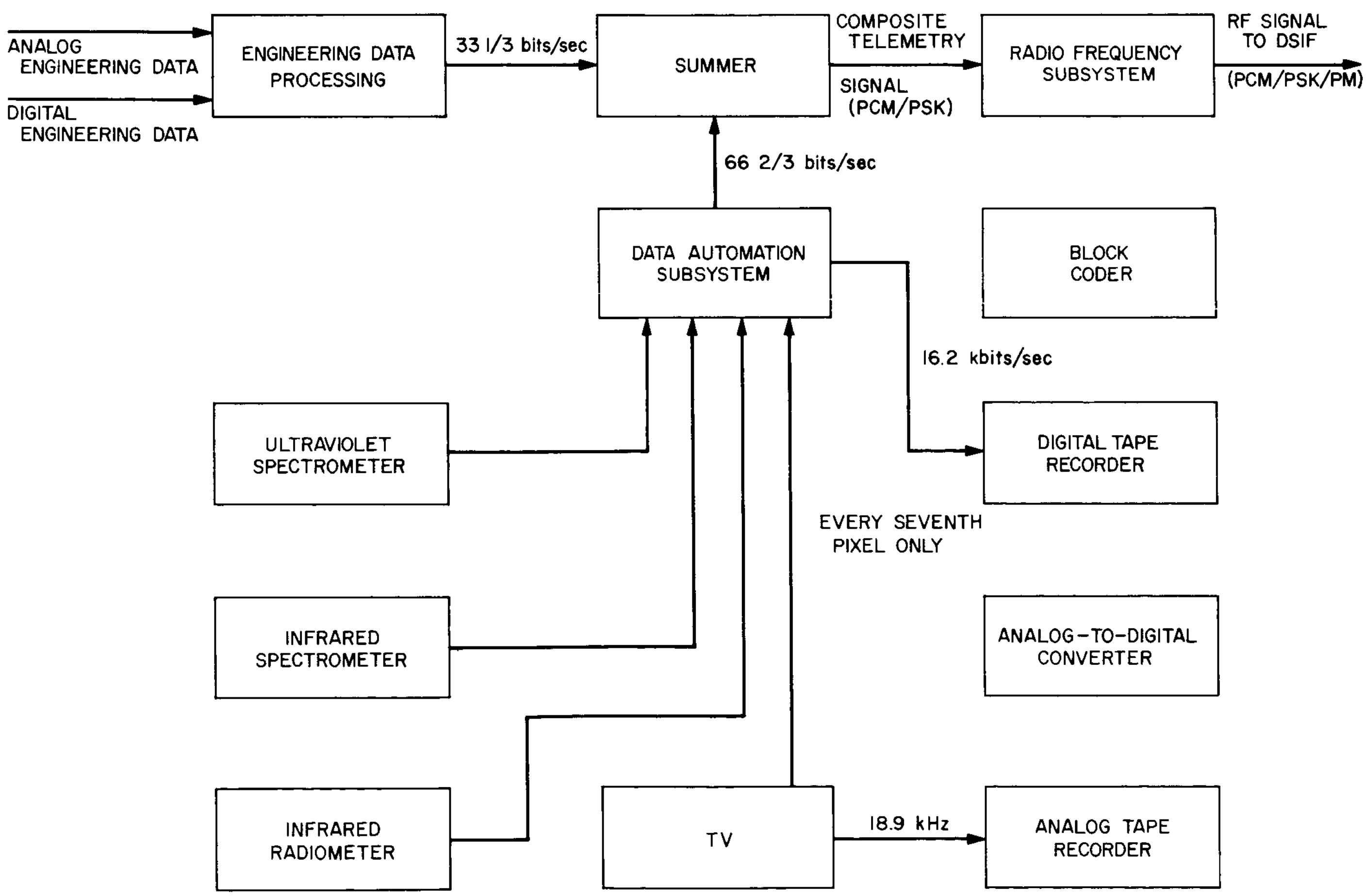


Fig. 1. Data flow during nominal encounter mode

IV. DSN Capabilities and Facilities

THE DEEP SPACE NETWORK

Established by the NASA Office of Tracking and Data Acquisition and under the system management and technical direction of JPL, the Deep Space Network (DSN) is responsible for two-way communications with unmanned spacecraft traveling from approximately 10,000 mi from the earth to interplanetary distances. [Earth-orbiting scientific and communications satellites and the manned spacecraft of the *Gemini* and *Apollo* Projects are tracked by the Space Tracking and Data Acquisition Network (STADAN) and the Manned Space Flight Network (MSFN), respectively.] NASA space exploration projects supported, or to be supported, by the DSN include the following: (1) *Ranger, Surveyor, Mariner,* and *Voyager* Projects of JPL; (2) *Lunar Orbiter* Project of the Langley Research Center; (3) *Pioneer* Project of the Ames Research Center; and (4) *Apollo* Project of the Manned Spacecraft Center (as backup to MSFN).

Present DSN facilities permit simultaneous control of a newly launched spacecraft and a second spacecraft already in flight. In preparation for increased U.S. activities in space, a capability is being developed for simultaneous control of either two newly launched spacecraft plus two in flight, or four spacecraft in flight. Advanced communications techniques are being implemented to enable obtaining data from, and tracking spacecraft to, planets as far out in space as Jupiter. The main elements of the DSN are described below.

A. Deep Space Instrumentation Facility

The Deep Space Instrumentation Facility (DSIF) is composed of tracking and data acquisition stations around the world. The deep space stations (DSSs) of the

DSIF and the deep space communication complexes (DSCCs) they comprise are as follows:

DSS	DSCC
Pioneer Echo Venus Mars	Goldstone
Woomera Tidbinbilla Booroomba[a]	Canberra
Johannesburg	
Robledo Cebreros Rio Cofio[a]	Madrid
Cape Kennedy (spacecraft monitoring)	
Ascension Island (spacecraft guidance and command)	

[a] Station not yet authorized.

These stations are situated such that three may be selected approximately 120 deg apart in longitude in order that a spacecraft in or near the ecliptic plane is always within the field of view of at least one of the selected ground antennas. JPL operates the U.S. stations and the Ascension Island DSS. The overseas stations are normally staffed and operated by government agencies of the respective countries, with the assistance of U.S. support personnel.

The Cape Kennedy DSS supports spacecraft final checkout prior to launch, verifies compatibility between the DSN and the flight spacecraft, measures spacecraft frequencies during countdown, and provides telemetry reception from liftoff to local horizon. The other stations obtain angular position, velocity (doppler), and distance (range) data for the spacecraft and provide command control to (uplink) and data reception from (downlink) the spacecraft. Large antennas, low-noise phase-lock receiving systems, and high-power transmitters are utilized. The 85-ft-diameter antennas have gains of 53 db at 2300 MHz, with a system temperature of 55°K, making possible significant data rates at distances as far as the planet Mars. To improve the data rate and distance capability, a 210-ft-diameter antenna was built at the Mars DSS, and two additional antennas of this size are planned for installation at overseas stations. In their present configuration, all stations except the Johannesburg DSS are full S-band stations. The Johannesburg DSS now has a GSDS S-band receiver-exciter subsystem.

It is the policy of the DSN to continuously conduct research and development of new components and systems and to engineer them into the network to maintain a state-of-the-art capability. Therefore, the Goldstone DSCC is also used for extensive investigation of space tracking and telecommunications techniques, establishment of DSIF/spacecraft compatibility, and development of new DSIF hardware and software. New DSIF equipment is installed and tested at the Goldstone DSCC before being accepted for system-wide integration into the DSIF. After acceptance for general use, the equipment is classed as Goldstone Duplicate Standard in order to standardize the design and operation of identical items throughout the system.

B. *Ground Communication System*

To enable communications between all elements of the DSN, the Ground Communication System (GCS) uses voice, teletype, and high-speed data circuits provided by the worldwide NASA Communications Network between each overseas deep space station, the Cape Kennedy DSS, and the Space Flight Operations Facility (SFOF, described below). The NASA Communications Network is a global network consisting of more than 100,000 route mi and 450,000 circuit mi interconnecting 89 stations, of which 34 are overseas in 18 foreign countries. Entirely operationally oriented, it is comprised of those circuits, terminals, and pieces of switching equipment interconnecting tracking and data acquisition stations with, for example, mission control, project control, and computing centers. Circuits used exclusively for administrative purposes are not included.

Voice, teletype, high-speed data, and video circuits between the SFOF and the Goldstone DSCC are provided by a DSN microwave link.

C. Space Flight Operations Facility

During the support of a spacecraft, the entire DSN operation is controlled by the Space Flight Operations Facility (SFOF) at JPL. All spacecraft command, data processing, and data analysis can be accomplished within this facility. Operations control consoles, status and operations displays, computers, and data processing equipment are used for the analysis of spacecraft performance and space science experiments. Communications facilities are used to control space flight operations by generating trajectories and orbits and command and control data from tracking and telemetry data received from the DSIF in near-real time. The telemetry, tracking, command, and station performance data recorded by the DSIF are also reduced at the SFOF into engineering and scientific information for analysis and use by scientific experimenters and spacecraft engineers.

V. DSIF Development and Operations

THE DEEP SPACE NETWORK

A. Flight Project Support

1. *Surveyor* Project

The *Surveyor III* spacecraft was launched from Cape Kennedy, Florida, on April 17, 1967. The Pioneer DSS provided primary tracking and command data transmissions. The Echo DSS performed backup tracking during first view periods, midcourse maneuvers, and landing. After *Surveyor III* landing on the moon, the Pioneer DSS continued in command while the spacecraft performed soil sampling and photographic experiments.

2. *Mariner* Mars 1964 Project

Early in March the *Mariner IV* spacecraft was tracked for two view periods, with the Venus DSS assisting the Mars DSS. After the Mars DSS tracking was halted for improvement work, all *Mariner IV* tracking was stopped because the spacecraft signal strength was below the receiver threshold of all other DSIF antennas. However, spacecraft signal strength is increasing as the *Mariner IV* approaches the earth for its prime conjunction later this year, and tracking will be resumed.

3. *Lunar Orbiter* Project

Lunar Orbiter II and *III* spacecraft tracking was continued, primarily on alternate occultation view periods. The video readout sequence of each spacecraft terminated, and the telemetry data received was of a normal cruise mode type. Simultaneous tracking of both spacecraft with the S-band receiver was also accomplished, both being tracked for approximately 1.5 hr with a doppler separation briefly approaching 75 Hz.

The Echo DSS and the Goldstone DSCC Apollo station successfully performed handover experiments with a test model of the *Lunar Orbiter* spacecraft (Fig. 1). The test model was installed at the Goldstone Test Facility screenroom and a passive link antenna was installed between Echo, Goldstone Apollo, and Pioneer MSFN stations. Adequate two-station testing of handover procedures and station equipment was obtained.

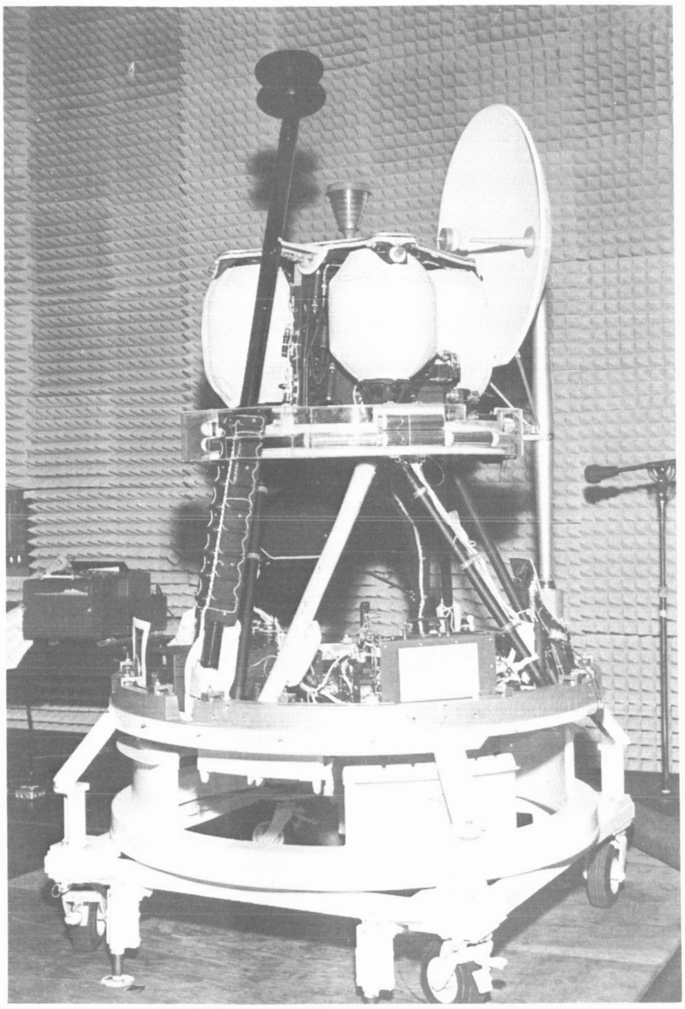


Fig. 1. *Lunar Orbiter* test model

The first launch readiness dates for *Lunar Orbiter IV* are scheduled for early May 1967. Unlike previous *Lunar Orbiter* missions, the primary objective of this mission will be to photograph 85% of the front side of the moon at resolutions between 50 and 100 m. To do this, a high-altitude lunar orbit will be used in conjunction with the high-resolution (24 in.) telephoto lens. There will be no preliminary orbit; the deboost maneuver will inject the spacecraft directly into the final picture-taking orbit.

The inertial attitude of the *Lunar Orbiter IV* orbit results in two phenomena that did not occur for the first three *Lunar Orbiters*. Because the sun–moon line will be almost perpendicular to the spacecraft lunar orbit early in the mission, sun occultation will not occur until two or three months of orbital flight. The spacecraft will be in sunlight during this whole time. Another characteristic of this trajectory is the absence of earth occultation for the first ten days of lunar orbit. Continuous tracking may be possible, contingent on tracking station availability.

4. *Pioneer* Project

Pioneer VII spacecraft tracking continues. When the Mars DSS became unavailable for continuous tracking because of azimuth bearing improvement work, the Echo DSS assumed temporary prime tracking duties. With the spacecraft over 45 million miles from earth, the signal received at the Echo DSS was at approximately ground receiver threshold.

B. DSS Equipment Installation and Testing

1. Pioneer DSS

Installation of two new generators at the Pioneer DSS is essentially completed. The generators provided primary power for operations, and for the S-band system, during the period from *Surveyor III* launch to 5 days after the landing on the moon.

2. Echo DSS

Installation of the *Mariner* Venus 67 read-write-verify command equipment and the ground telemetry system is completed. System integration testing between the Echo DSS, the Mars DSS, and the SFOF is scheduled for early May.

For *Pioneer VII* tracking, two equipment modifications were made at the Echo DSS to reduce the telemetry bit error rate. First, a waveguide section was installed in the cone assembly. It was connected between the flange connection of the righthand circular polarization sum port and the input port to the 26-db directional coupler so as to bypass the S-band-Cassegrain-monopulse sum-channel polarize select and the diplexer. Approximately a 1.75-db lower receiver threshold was achieved; however, the modification limited the system to a receive-only configuration. A second modification was made to reduce the receiver reference loop bandwidth to 3 Hz and to lower the ground receiver threshold approximately 5 db. The resulting signal-to-noise improvement was sufficient to allow the Echo DSS to track the *Pioneer VII* spacecraft.

3. Venus DSS

The 30-ft az-el antenna at the Venus DSS was used for a time synchronization experiment. This experiment was successfully conducted between the Venus and Echo DSSs, and between the Venus DSS and JPL, Pasadena, using the moon as a reflector.

The 85-ft az-el antenna at the Venus DSS is being used for a monostatic planetary radar experiment. This experiment, with Mars as the target, is in progress at S-band (2388 MHz).

4. Mars DSS

Tracking operations at the Mars DSS were suspended because of azimuth bearing improvement work. Testing and installation of the *Mariner* Venus 67 backup read-write-verify command subsystem was completed. The S-band system underwent a thorough testing in preparation for backup tracking of the *Surveyor III* landing on the moon.

C. Communications Development and Testing

1. Multiple Mission Support Capability

Multiple mission support is a new operating concept for the DSIF. This new concept provides an increase in DSS efficiency and flexibility and improves the quality and reliability of flight project support.

The design of the multiple mission support concept allows a station to support, simultaneously, two or more projects, one tracking and the others testing. In addition, for those DSSs located in a DSCC having microwave facilities between stations, the utilization of more than one station to support two flight projects located at a single station can be realized.

Implementation of the multiple mission support concept creates three distinct and separate areas: (1) the mission independent area, which houses the basic tracking station equipment required to support any flight project and is independent of the flight project, (2) the mission support area, which houses the flight-project-furnished equipment (outside the control room), and (3) the multiple mission support area, which houses the standard equipment supplied by the DSIF for supporting and interfacing with each flight project.

The multiple mission support area is being implemented in two phases:

(1) Phase I provides multiple mission support capability at the Pioneer and Robledo DSSs for the *Surveyor* and *Mariner* Venus 67 projects, and at Echo DSS for the *Lunar Orbiter* and *Pioneer* projects. Phase I configurations at the Mars and Cebreros DSSs will be limited to aiding nearby stations to fulfill multiple mission support capability via the available microwave links. A two-station multiple support configuration is shown in Fig. 2.

(2) Phase II will provide the DSIF with a permanent multiple mission support capability.

2. Multi-Mission Telemetry Demodulation System

Since its inception, the DSN has carried, and will continue to carry, an ever-increasing load of spacecraft tracking commitments. Each of these various spacecraft has its own particular requirements in the demodulation and decommutation of its telemetry signals. At present, these individual requirements are being met through the use of equipment specifically designed to handle one particular spacecraft or series of spacecraft. The continued use of this mission-dependent equipment is both costly and time consuming, not only in its fabrication, but in the time and space required to implement the DSN for each new spacecraft design. Thus, the need became increasingly clear for a general-purpose telemetry demodulation system that was truly mission-independent.

A program has been initiated to develop the methods and equipment necessary to implement a mission-independent telemetry demodulator. The phases of this program are:

(1) *Phase I.* The first goal of the program was to establish that telemetry data could be recovered by using only a digital computer as a subcarrier demodulator and detector. This goal was successfully achieved on an SDS 930 computer, programmed in Fortran, and operating upon the tape-recorded *Mariner IV* telemetry signals.

(2) *Phase II.* Since Phase I was a non-real-time experiment, the next step was to demonstrate that this same correlation technique could be used for real-time telemetry data recovery. This phase was accomplished by using an SDS 930 computer operating on *Mariner IV* telemetry signals as received on the Mars DSS antenna. The computer data output was appropriately formatted and sent over teletype lines to JPL, thus presenting a real-time telemetry data printout.

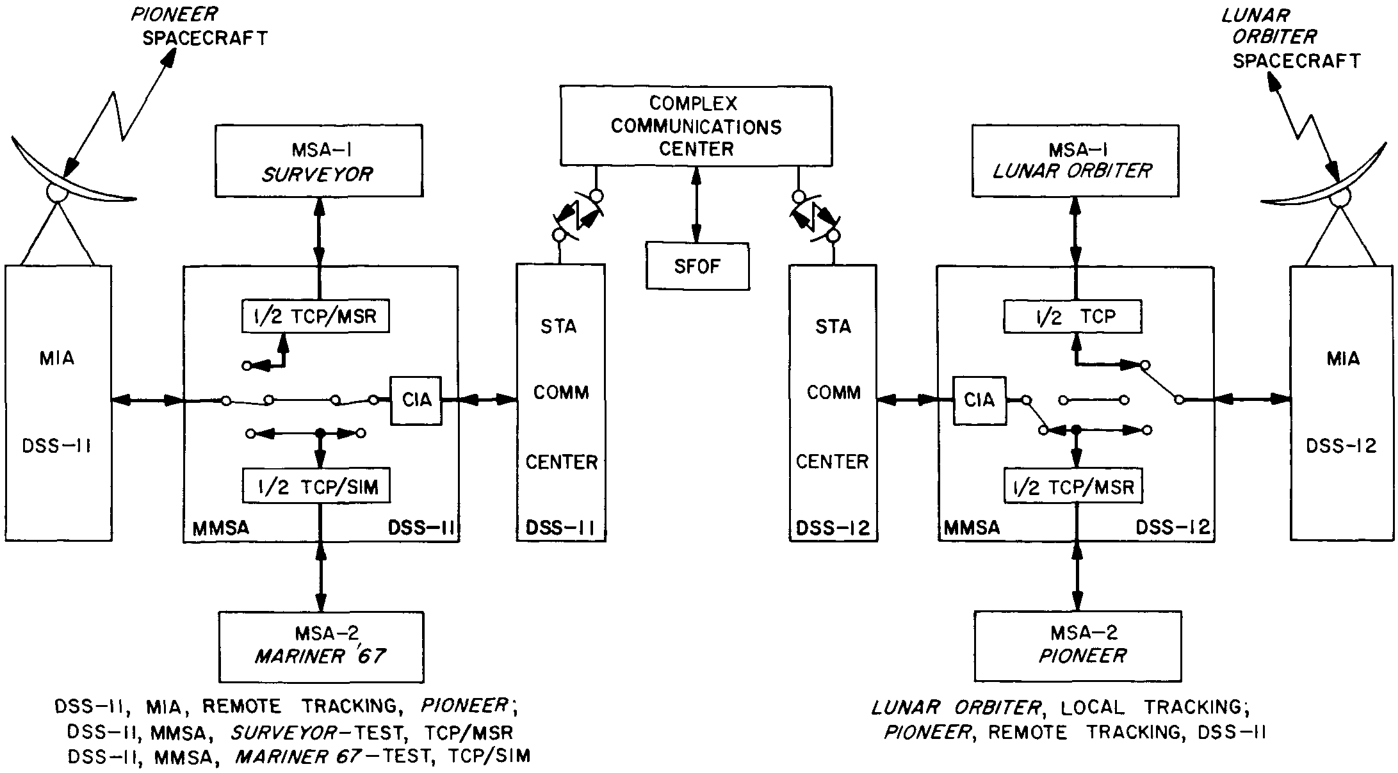


Fig. 2. Two-station multiple mission support configuration

(3) *Phase III.* While the first two phases of the program had been primarily a demonstration of concept, Phase III was the first attempt to implement a system which was a reasonable analog of the proposed operational system. Hence, the Phase III or Type III demodulator was constrained to use an SDS 920 computer (the 920 is one-fifth as fast as the 930 computer) and to do certain operations in external general-purpose demodulation hardware. The external hardware performs the subcarrier demodulation and data stream integration, and the computer performs the detection and the data-dependent controlling operations.

(4) *Phase IV.* Phase IV will be the development of the actual operational system.

3. R&D Receiver Subsystem

A new R&D receiver subsystem completely compatible with present DSIF receiver subsystems has been built and evaluated as part of the *Mariner* Venus 67 ranging system (Fig. 3). This subsystem provides the R&D receiver designer and experimenter with complete flexibility regarding module location, module addition, or module removal within the subsystem.

The universal R&D receiver housing subsystem differs from the standard DSIF subsystem in that complete mechanical and electrical module mounting flexibility is provided. The module mounting plate is directly interchangeable with the DSIF standard, but it has been redesigned to provide three vertical slots with a standard pattern of tapped holes. A family of module adapter plates and blank filler plates has been designed to fit into the slots as the system design dictates. The adapter plates convert standard DSIF receiver components to a modular configuration. The adapter plates also contain a mounting support for the dc power, module control, and monitoring connector and cable.

To provide the degree of flexibility needed, the subsystem has a permanent signal, power, and monitor cabling system between the frames and the cabinet. Each frame has 42 coaxial cables which originate at a jack field on the cabinet side, and which terminate on the rear of the frame. The coaxial cables from the modules to the frames are run in a plastic duct which eliminates time needed to replace cables when modules are added or removed. Each frame has 64 dc control and monitoring cables that originate in a junction box at the rear of the cabinet. All connections within the junction box are made

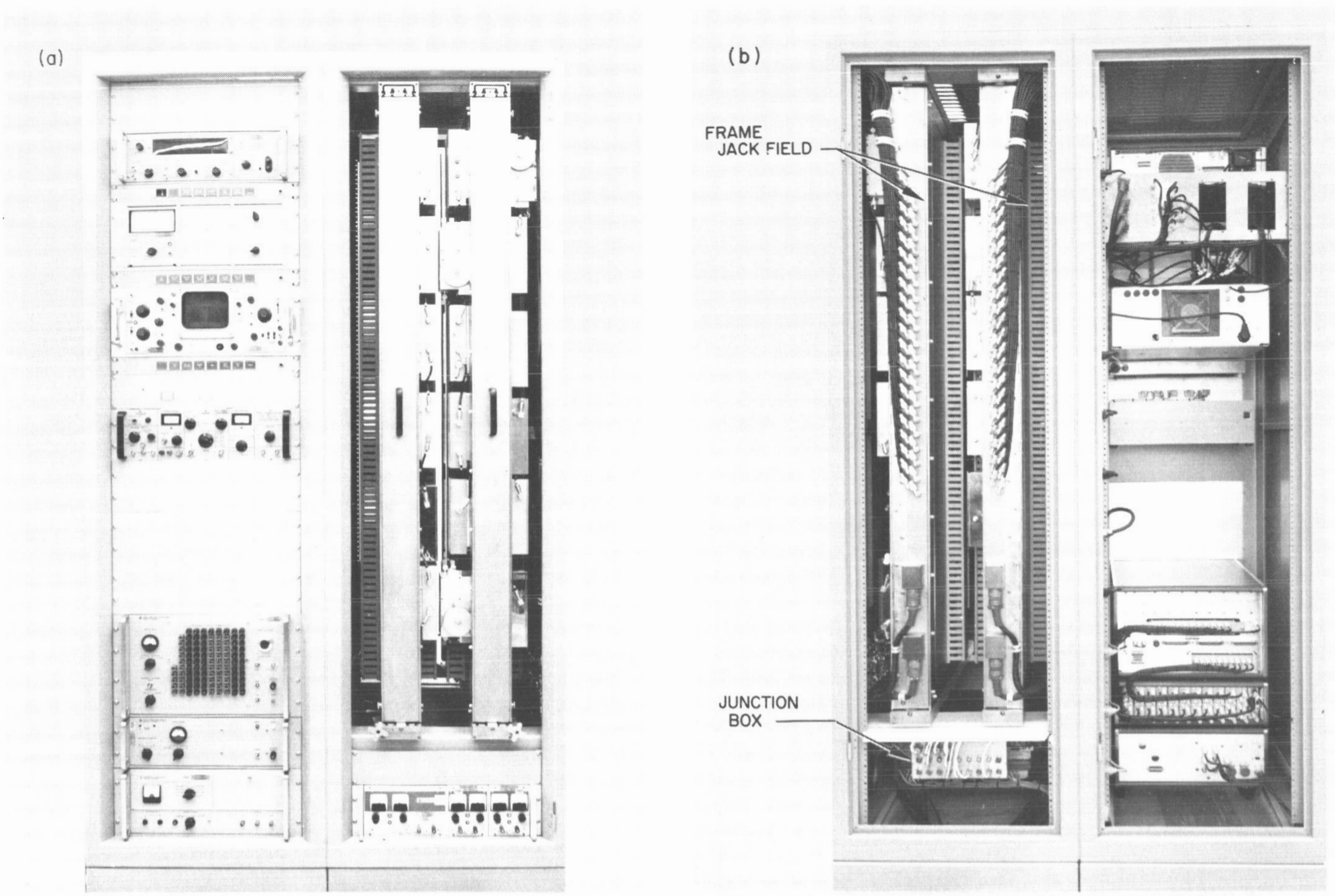


Fig. 3. *Mariner* Venus ranging system: (a) front view; (b) rear view

with taper pins inserted into terminal blocks. The junction box is used as a patchboard. The dc power, control, and monitoring cables terminate in terminal blocks located on the module mounting plates. To connect a new module into this system, a power and control cable is attached to an adapter plate, and the open end is inserted into the appropriate socket of the terminal block; then connections to remote locations are made at the junction box.

The universal R&D receiver housing subsystem has been proven in a first prototype destined for experimental use at the Mars DSS. A second subsystem is now in procurement to be used for advanced development projects at the Venus DSS.

4. Time Synchronization System

The precision time synchronization system (Fig. 4) was described in SPS 37-43, Vol. VI, pp. 20–21. Since that time, system tests have been performed using the 10-kw X-band transmitter and the 30-ft-diameter antenna at the Venus DSS; also, the receiver was installed in the communications building at the Echo DSS and its performance was tested. No significant problems were encountered during the system tests, and the results are in good agreement with the predicted system performance. The receiver performed as expected and will need only a minimum of changes before the first commercially built prototype is completed.

The performance of each of the segments of the time synchronization program was determined: (1) the program points the antenna at the ephemeris-predicted location of the moon to within ± 0.015 deg throughout an entire track and to within ± 0.006 deg throughout 80% of the track, (2) the program drives the frequency of the programmed oscillator to the ephemeris-determined value

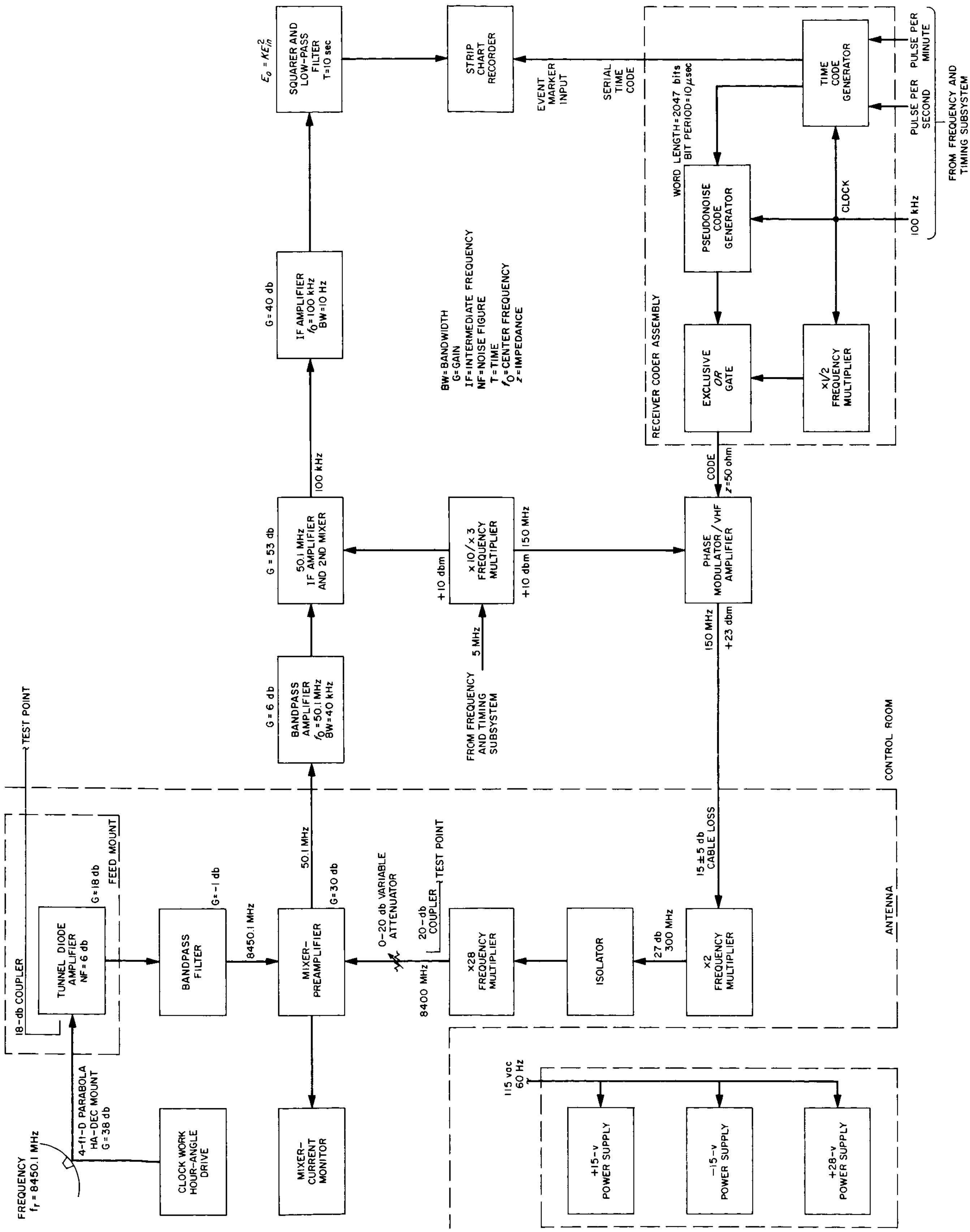


Fig. 4. X-band time synchronization block diagram

so well that the error cannot be detected with the equipment available at the DSS, and (3) the coder control by the program is well within the accuracy requirements of the system.

Only one factor remains to be checked: that the scattering properties of the moon are essentially the same when the receiver and transmitter are widely separated on the earth as they are when the receiver and transmitter are very close. To check this factor, the receiver will be installed at the Tidbinbilla DSS in Australia in the near future.

5. Telemetry Command Processor

The telemetry command processor (phase II) provides the DSIF with a mission-independent telemetry and command data processing capability for real-time operation. The telemetry command processor assembly interfaces with the mission-dependent project equipment and provides the capability to decommutate, process, edit, alarm-monitor, and format spacecraft telemetry data for transmission to the SFOF, and to process and verify command data, received at the DSS from the SFOF, for transmission to the spacecraft.

Basic telemetry command processor (phase II) implementation, with a modification consisting of the addition of a fourth teletype send module, has been completed at all DSSs. The modification establishes a teletype communications channel for transmitting spacecraft automatic-gain-control and static-phase-error data between the telemetry command processor and the data display system. The telemetry command processor (phase II) configuration is being expanded to: (1) increase capabilities to digital recording of station data, and (2) process support to mission-independent telemetry demodulation.

6. Error-Correcting Encoder-Decoder for DSN Teletype

An error-correcting encoder-decoder was demonstrated on DSN teletype lines. Employment of this error-correction system ensures utmost reliability when spacecraft commands generated in the SFOF at JPL are transmitted to the deep space stations. The system uses an SDS 910 computer as a prototype communications processor, together with an associated communications buffer. Thirty-six bits are encoded into 60 bits; each teletype character corresponds to a 4-bit symbol. The system is capable of correcting up to 2 symbol errors in a code word of 15 symbols, and detecting up to 4 symbol errors. In addition, synchronization errors caused by insertions or deletions of characters are detected. The system is compatible with NASCOM constraints, and has been demonstrated over NASCOM lines. The undetected error probability per coded word using this system is about 0.5×10^{-8} in the worst (noisiest) case; the probability of a detected but uncorrected error (including sync errors) is about 1 in 200 in the worst case.